Ion Propulsion for a Mars Sample Return Mission

John R. Brophy* and David H. Rodgers**
Jet Propulsion Laboratory
California Institute of Technology
Pasadena, California

Improvements to the ion propulsion system that flew are Deep Space 1 are currently being developed for use on a variety of near-term, flag-ship, deep-space science missions. This same improved system could be used for Mars sample return missions to reduce risk. A study was conducted to examine the feasibility of using SEP to perform a Mars sample return mission from a single medium-class launch vehicle. This study concluded that the medium-class Atlas V – 531 launch vehicle, together with an advanced, near-term SEP system could deliver an 1800-kg lander to Mars and return the samples to Earth. In addition, the use of SEP eliminates the need for aerobraking and aerocapture significantly reducing the overall mission risk. The use of SEP also enables access to the entire Martian surface at every launch opportunity.

Introduction

With the successful flight of the ion propulsion system on Deep Space 1 (DS1) solar electric propulsion (SEP) has now entered the mainstream of propulsion options available for deep-space missions [1,2,3]. Several scientifically interesting deep-space missions are now looking at the use of ion propulsion to significantly reduce total mission costs. These missions include Comet Nucleus Sample Return (CNSR), Venus Surface Sample Return (VSSR), Saturn Ring Observer, Titan Explorer, Neptune Orbiter, and Europa Lander.

This paper describes an on-going study to examine the feasibility of using SEP based on derivatives of the Deep Space 1 ion propulsion system (IPS) technology to enable a Mars Sample Return (MSR) mission to be performed from a single, medium-class launch vehicle. In addition, this paper lists the other mission and science benefits enabled through the use of SEP that have been identified in this study.

*NSTAR Project Manager, Member AIAA
**Deputy Director for the Center for In Situ Science and Sample Return

IPS Technology for MSR

The DS1 IPS was provided by the NASA Solar electric propulsion Technology Application Readiness (NSTAR) project [4]. A trade study performed in support of the CNSR advanced mission study activity identified improvements to the NSTAR ion propulsion technology that could be developed to provide significant mission benefits to CNSR without incurring unacceptable technical risks [5]. These improvements include increasing the maximum engine specific impulse from 3100 seconds to 3800 seconds, increasing the maximum engine input power from 2.3 kW to 3.1 kW, and increasing the engine propellant throughput capability from 88 kg to 195 kg.

For the Mars sample return mission study using SEP described herein it was decided to assume the use of these advanced capabilities since a technology program was in place to make these improvements. In addition, it was decided that the overall SEP system for this study should be as nearly identical as possible to that identified for the CNSR mission in order to minimize the non-recurring costs for the IPS hardware, systems engineering and the solar array. Consequently, the MSR study assumed the use of the same sized solar array that was being investigated for CNSR. This array consists of two wings with a total power of 17 kW beginning-of-life (BOL), at a distance of 1 AU from the sun (each wing is 8.5 kW).

The complete SEP system includes a total of four advanced NSTAR ion engines, four upgraded power processor units (PPUs), a xenon feed system (XFS), and two digital control and interface units (DCIUs). The PPUs are assumed to be upgraded from the NSTAR technology so that they can process a maximum output power of 3.1 kW instead of the NSTAR maximum of 2.3 kW. This is accomplished by adding an additional module to the high-voltage beam power supply to increase its maximum output voltage capability from 1200 V to 1500 V. The maximum input power to the IPS is 10 kW. Under normal operation all four engine-PPU strings are operated simultaneously with a PPU input power of 2.5 kW. In the event of a failure of one engine or one PPU the remaining three engine-PPU strings are operated with a PPU input power of 3.4 kW. This results in the same system thrust level and the spacecraft will fly the same trajectory. Thus, the IPS can tolerate a single failure of an engine, PPU or DCIU without affecting the mission. Furthermore, this single-fault-tolerance capability comes without having to pay the mass penalty of having to add an additional engine-PPU string and its associated support equipment (structure, gimbal, plumbing, cabling, and thermal control). The PPUs are assumed to be cross-strapped to the thrusters as shown in Fig. 1.

MSR SEP Mission Design

To take the best advantage of the on-board SEP system it is desirable to use it for as much of the mission as possible. With this in mind, the following mission scenario was selected. The MSR spacecraft consists of a lander and a carrier vehicle which includes the SEP system. The lander is assumed, for the purposes of this study, to be the '05 lander design in terms of mass and volume [6]. During launch and the cruise to Mars the lander and carrier vehicle are matted together with the lander sitting on top of the carrier vehicle in the launch configuration as shown in Fig. 2.

The lander mass is approximately 1700 kg, which must be supported by the carrier vehicle during launch. To minimize the required structure mass, the carrier vehicle was made as short as possible. To provide sufficient surface area for the PPU radiators and the spacecraft avionics and power system, the carrier vehicle width was increased as its height was decreased. A comparison of the MSR SEP carrier vehicle with the CNSR SEP vehicle is given in Fig. 3

The launch vehicle takes the combined lander-carrier vehicle to Earth escape with a slightly positive hyperbolic excess (C_3) energy. The SEP system is used to complete the heliocentric transfer to Mars and places the combined vehicle into a 5-day elliptical orbit around Mars. The cruise configuration of the combined lander-carrier vehicle is given in Fig. 4. The transfer trajectory to Mars is shown in Fig. 5 for an initial C_3 of 0.6 km^2/s^2.

The low-thrust SEP system is capable of providing the delta-V necessary to get into Mars orbit and the mechanics of this transfer have recently been worked out by Sweetser at JPL [7]. The capture and 5-day elliptical orbits at Mars are shown in Fig. 6. The lander is released from the carrier in this elliptical orbit and uses a small delta-V (of order 6 m/s) to deorbit and subsequently lands on the Martian surface.

While the lander is on the surface of Mars, the carrier vehicle is using the SEP system to spiral down to a 500-km altitude circular orbit around Mars as suggested in Fig. 6. After the lander has collect the Martian samples the Mars Ascent Vehicle (MAV) lifts them to low Mars orbit. For this study it was assumed that the delta-V of up to 100 m/s required for the carrier vehicle to rendezvous with the samples is accomplished with an on-board hydrazine propulsion system. A better approach perhaps would be to use the SEP system to provide most of this delta-V. The high specific impulse of the ion propulsion system should enable the carrier vehicle to accommodate much larger differences between the carrier's orbit and the orbit into which the MAV places the samples.

After collecting the Mars samples, the carrier vehicle uses the SEP system to spiral out from low Mars orbit to an escape trajectory. Finally, the SEP system puts the spacecraft on a direct Earth entry trajectory, where the Earth entry vehicle returns the samples to Earth's surface. After the Earth entry vehicle is separated from the carrier vehicle the SEP vehicle is diverted away from the Earth entry trajectory using the on-board hydrazine propulsion system to provide a divert delta-V of 40 m/s. The Earth return trajectory is shown in Fig. 7. The total delta-V provided by the SEP system for this mission is approximately 11,500 m/s. The breakdown by mission phase is given in Table 1.

Table 1 Low-Thrust Delta-V Summary

Mission Phase	Delta-V (m/s)
Earth-Mars transfer and capture to 5-day Mars elliptical orbit	2600
Spiral to low Mars orbit	2800
Spiral to Mars escape	2900
Mars-Earth transfer	3150
Total	11450

Refinements to this overall scenario include evaluation of using the SEP system to provide most of the sample rendezvous delta-V and using the SEP system to return the samples to low Earth orbit. Returning to low Earth orbit using the SEP system may have planetary protection advantages relative to direct entry.

SEP System Design

The SEP system mass breakdown is given in Table 2. This table assumes a 5-engine system. This system has one more engine than originally planned because of the large propellant load. With a propellant load of order 800 kg of xenon, at least four engines are required to process this even assuming the individual engine throughput capability has been increased from 88 kg to 200 kg of xenon. The fifth thruster is added to maintain the single-fault-tolerance capability. The system still only includes four PPUs since the PPUs should not be subject to wear-out failures provided the thermal constraints governing their use are maintained.

The tankage mass fraction is 5% of the total propellant mass stored. The total propellant load is that calculated by from the trajectory analysis plus 10% which includes 3% residuals and 7% to cover flow control tolerance, startup, and other miscellaneous affects. The trajectory analysis assumed a 90% SEP duty cycle during SEP operation and end-of-life engine performance for the entire mission.

Vehicle Mass Summary

A mass breakdown of the combined lander and carrier vehicle is given in Table 3. A growth contingency of 30% has been added to all items except the Sample Capture System and the Lander. The designers of these systems quoted growth contingencies which were used as is in this study. The carrier dry mass (including growth) is 1115 kg and the total vehicle wet mass (including the lander) is 3794 kg. The launch vehicle selected is the Atlas V- 531. This is a medium-class launch vehicle with an injected mass capability of 5900 kg to a C_3 of 0.6 km^2/s^2. For the purposes of this study, the launch vehicle capability was derated by 10% to get 5310 kg. The difference between the derated launch vehicle capability and the total spacecraft wet mass is 1520 kg which results in a launch vehicle margin of 28%. The total xenon propellant load is 780 kg including 10% contingency.

Benefits of SEP for MSR

The use of SEP for a Mars sample return mission according to the scenario described above has several attractive advantages. First, this approach enables a sample return mission using a single medium-class launch vehicle. Second, the use of SEP significantly relaxes the constraint on launch periods, where the launch period may be extended to one or two months with little mass penalty.

Third, the low-thrust SEP trajectory to Mars ends with the vehicle arriving at Mars with a nearly zero relative velocity. This allows the spacecraft to enter into nearly any desired orbit inclination at Mars. This, in turn, allows the lander to be targeted to any position on the Martian surface at every Earth-launch opportunity.

Fourth, the nearly zero relative arrival velocity of the SEP trajectory eliminates the need for aerocapture at Mars. In addition, the use of the SEP system to spiral down to a low Mars orbit eliminates the need for aerobraking. The elimination of both aerocapture and aerobraking significantly reduce the overall mission risk.

Fifth, because the SEP system is thrusting for a large fraction of the trip to Mars, it can be used to retarget lander post launch if updated information becomes available that makes retargeting highly desirable. Even after entering the 5-day elliptical orbit at Mars the SEP system could be used to provide some amount of retargeting for the lander prior to separation.

Finally, by carrying lander into an elliptical orbit around Mars, the landing can be delayed if necessary until conditions are favorable. This would provide, for example, the ability to wait out dust storms before landing.

Acknowledgements

The authors acknowledge the significant contributions to this work by the members of the SEP for MSR team including Frank Carsey, Keith English, Lloyd French, Marty Gilmore, Emily Eelkema, Jeff Hall, Don Hunter, Muriel Noca, Jon Sims, Ted Sweetser, and Brian Wilcox. The work described in this paper was conducted by the Jet Propulsion Laboratory, California Institute of Technology, under contract to the National Aeronautics and Space Administration.

Conclusions

The use of SEP for a Mars sample return mission enables the entire mission to be accomplished using a single medium-class launch vehicle (the Atlas V-531). The launch vehicle lifts a combined SEP carrier vehicle mated with a Mars lander to Earth escape with a slightly positive hyperbolic excess energy. The SEP system is used to carry the lander to Mars and place it in a 5-day elliptical orbit. From this orbit the lander separates from the carrier vehicle and de-orbits to land on Mars. The carrier vehicle uses the SEP system to spiral to low Mars orbit. The Mars ascent vehicle, which was part of the lander, lifts the Martian samples to low Mars orbit. The carrier vehicle rendezvous with the samples and uses the SEP system to escape from Mars and return to Earth. The Earth entry vehicle, with the samples, performs a direct entry at Earth atmosphere approximately 5 years after the start of the mission.

The use of SEP provides several other benefits including: eliminating the need for aerocapture and aerobraking; providing access to any point on the Martian surface at every launch opportunity; enabling the landing target to be changed after launch to respond to new information; and allowing the lander to wait until landing conditions on Mars are favorable.

References

1. Rayman, M. D., "Results from the Deep Space 1 Technology Validation Mission," IAA-99-IAA.11.2.01, presented at the 50th International Astronautical Congress, 4-8 Oct 1999, Amsterdam, The Netherlands.
2. Polk, J. E., et al., "In-Flight Performance of the NSTAR Ion Thruster Technology on the Deep Space One Mission," AIAA-99-2274, presented at the 35th AIAA/ASME/SAE/ASII Joint Propulsion Conference, June 21-23, 1999.
3. Brophy, J. R., et al., "Ion Propulsion," internal JPL document, presented at the Deep Space 1 Technology Validation Symposium, February 2000.
4. Christensen, J. A., et al., "The NSTAR Ion Propulsion Subsystem for DS1," AIAA-99-2972, presented at the 35th AIAA/ASME/SAE/ASII Joint Propulsion Conference, June 21-23, 1999.
5. Brophy. J. R., "Ion Propulsion System Design for the Comet Nucleus Sample Return Mission, AIAA-2000-3414, to be presented at the 36th AIAA/ASME/SAE/ASII, Joint Propulsion Conference, Huntsville, Al, July 17-19, 2000.
6. Roncoli, R., Navigation and Flight Mechanics Section, JPL, personal communication, May 2000.
7. Sweetser, T., Mission and Systems Architecture Section, JPL, personal communication, May 2000.

Table 2 SEP system mass breakdown

SEP System	QTY	CBE Unit Mass (kg)	CBE Total Mass (kg)	Comment
Engine	5	8.34	41.70	4 required to process 10.0 kW, may have to add one more for throughput
High Voltage and Neut. Supply Assembly	4	16.00	64.00	Internally redundant
DCIU	2	2.80	5.60	1 required plus one spare
Regulator	2	0.45	0.90	
Service Valve - HP	1	0.01	0.01	
Service Valve - LP	11	0.01	0.11	
Pressure Transducer	2	0.25	0.50	
Latch Valve - HP	2	0.10	0.20	6.65
Latch Valve - LP	5	0.10	0.50	
Filter	1	0.13	0.13	
Var. Reg.	15	0.15	2.25	
Tubing	1	1.75	1.75	0.5 kg per thruster * 4 thrusters = 2.0 kg
Fittings	1	0.30	0.30	0.1 kg per thruster * 4 thrusters = 0.4 kg
Gimbal	5	4.45	22.25	One for each thruster
Cabling	1	13.61	13.61	5% of dry mass
Xenon Tank	1	38.87	38.87	5% tankage mass fraction
Misc. Thermal Control	1	13.61	13.61	5% of dry mass
Structure	1	40.00	40.00	To support launch loads
		Total	***246***	

Table 3 MSR SEP Spacecraft Mass Summary

	Dry Launch Mass	
	Total CBE	CBE+ Uncer
Total Dry Mass at Launch (including lander)	2449	2902
Total Wet Mass at Launch (including lander)	3342	3794
Total Lander Mass	1584	1786
SEP Propellant	777	777
Chemical Propellant	116	116
Total Launch Mass	**3342**	**3794**
Orbiter (Sample Return Vehicle) - Dry Mass	**865**	**1115**
SEP Propulsion Dry Mass	**246**	**320**
Chemical Propulsion Dry Mass	**25**	**32**
Avionics and Power System	**45**	**58**
Power	**234**	**304**
Attitude Determination & Control (ADC)	**34**	**44**
Thermal	**20**	**26**
Telecommunications	**25**	**33**
Structures	**130**	**169**
Primary Structure	**98**	**128**
Solar Array Support (Structure, gimbals, etc)	**31**	**41**
Sample Capture System (for Rendevous)	**106**	**130**
Total Wet Lander Mass	**1584**	**1786**
Xenon Propellant	777	777
Hydrazine Propellant	116	116
Atlas V 531 Launch Capacity	5900	5310
Difference	**2558**	**1516**
remaining margin (reserve)	43.36%	28.54%

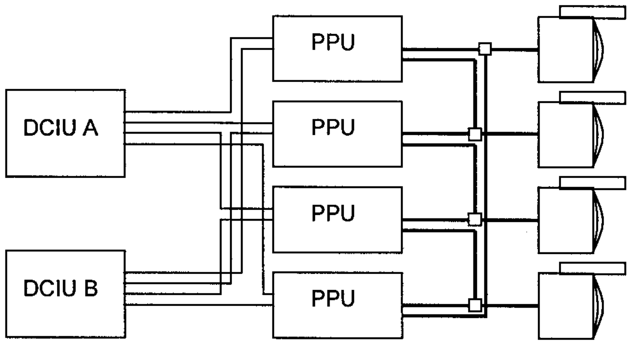


Fig. 1 IPS architecture with PPUs cross-strapped to the engines. Two DCIUs are included for redundancy.

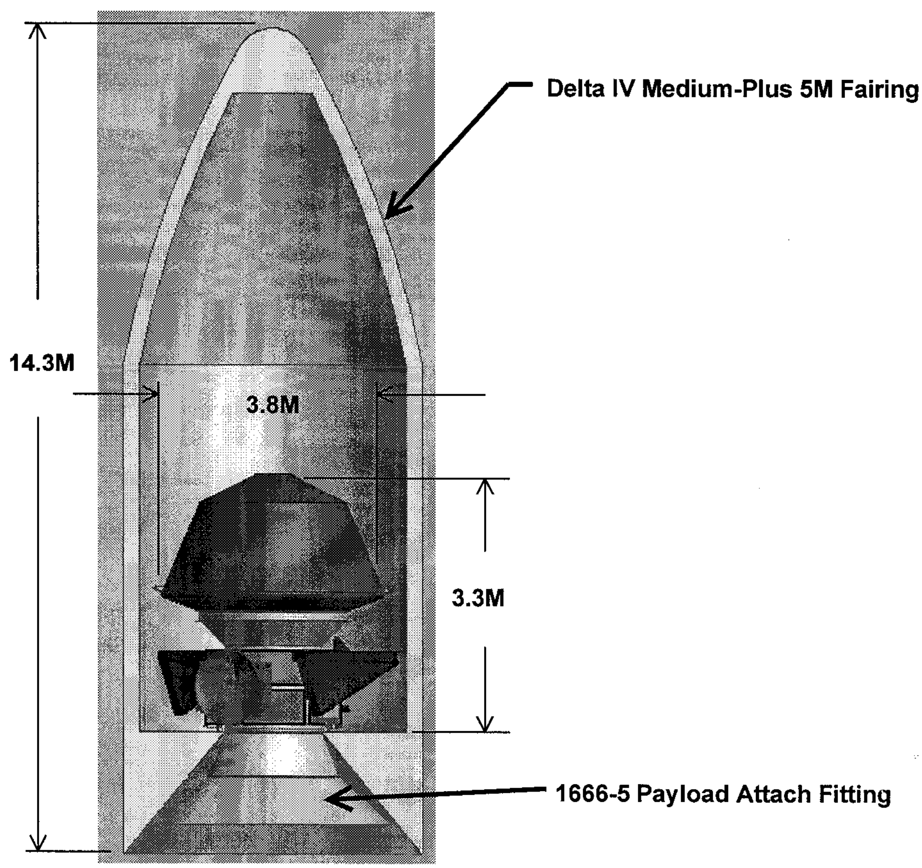
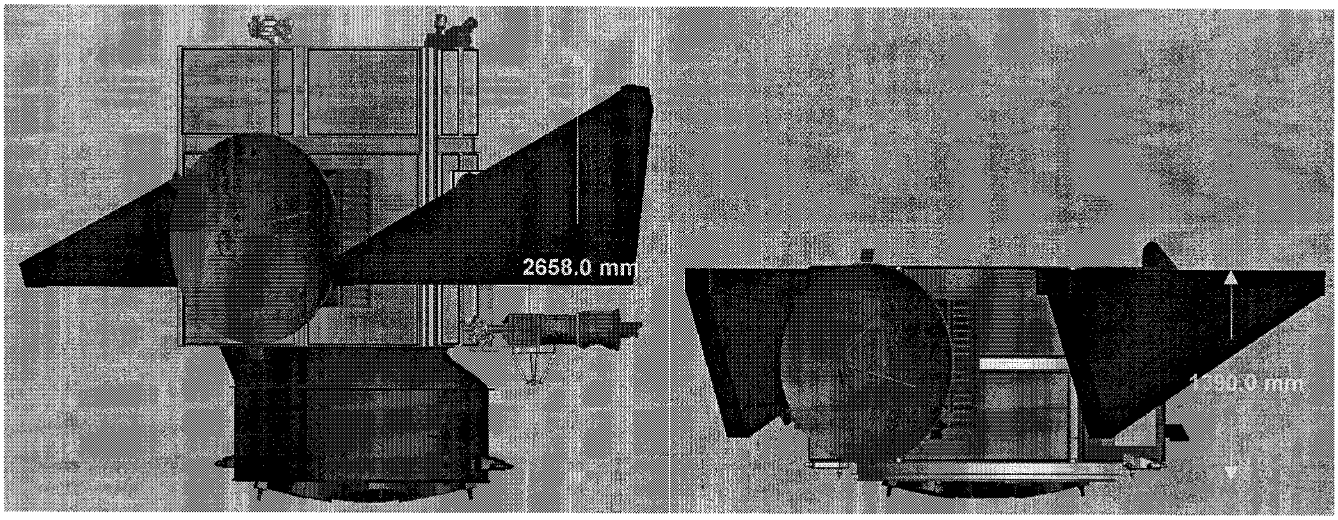


Fig. 2 MSR SEP vehicle in the launch vehicle fairing.



CNSR Bus **MSR SEP Bus**

Fig. 3 Size comparison of the CNSR SEP spacecraft bus with that of the MSR SEP carrier vehicle bus. The MSR SEP bus is much shorter to minimize the structure mass required to carry the heavy Mars lander.

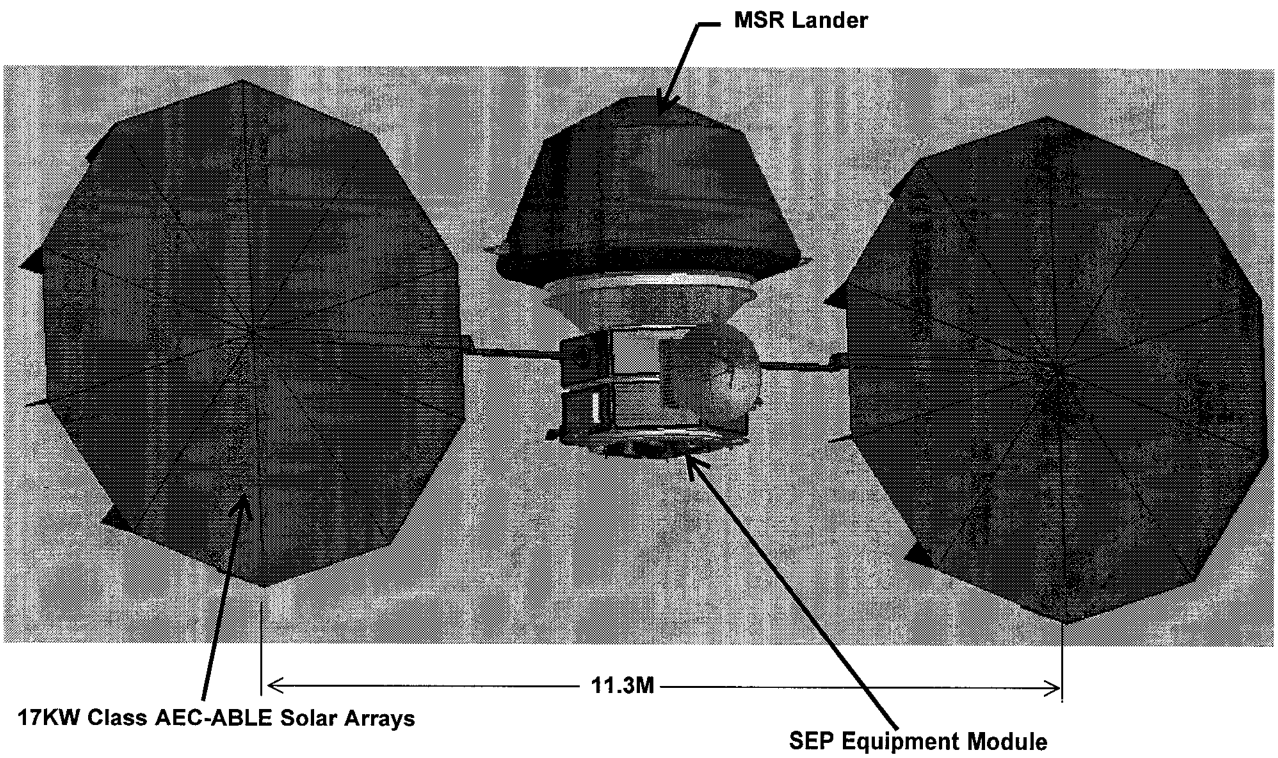


Fig. 4 Cruise configuration of the MSR SEP vehicle with the lander.

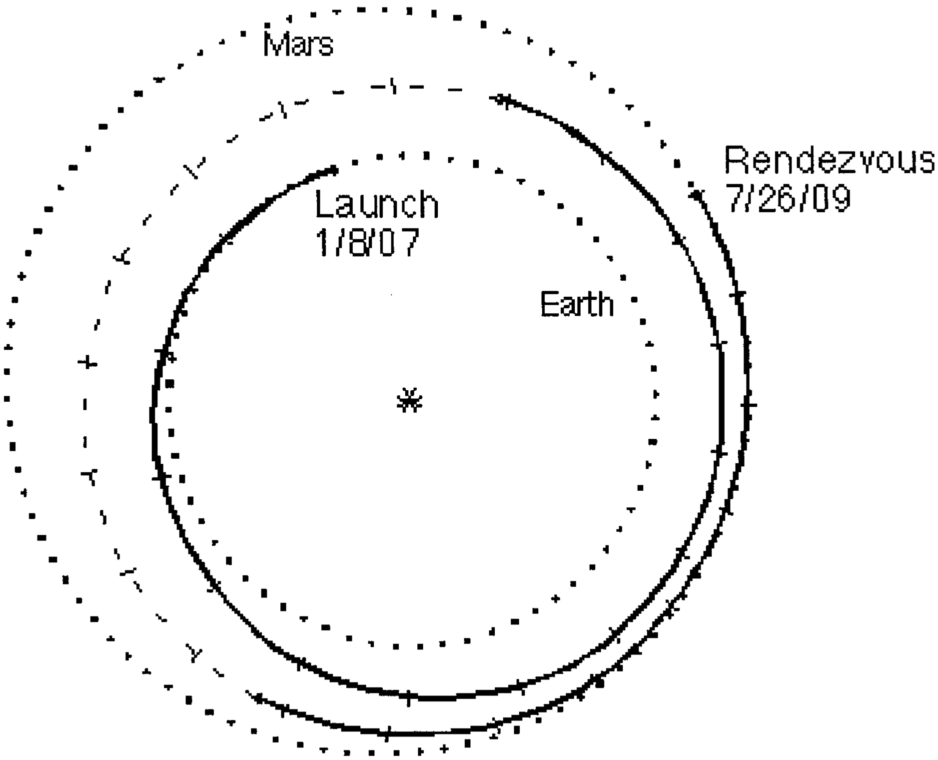


Fig. 5 Earth-to-Mars low-thrust trajectory beginning from an Earth-escape trajectory with $C_3 = 0.6\ m^2/s^2$.

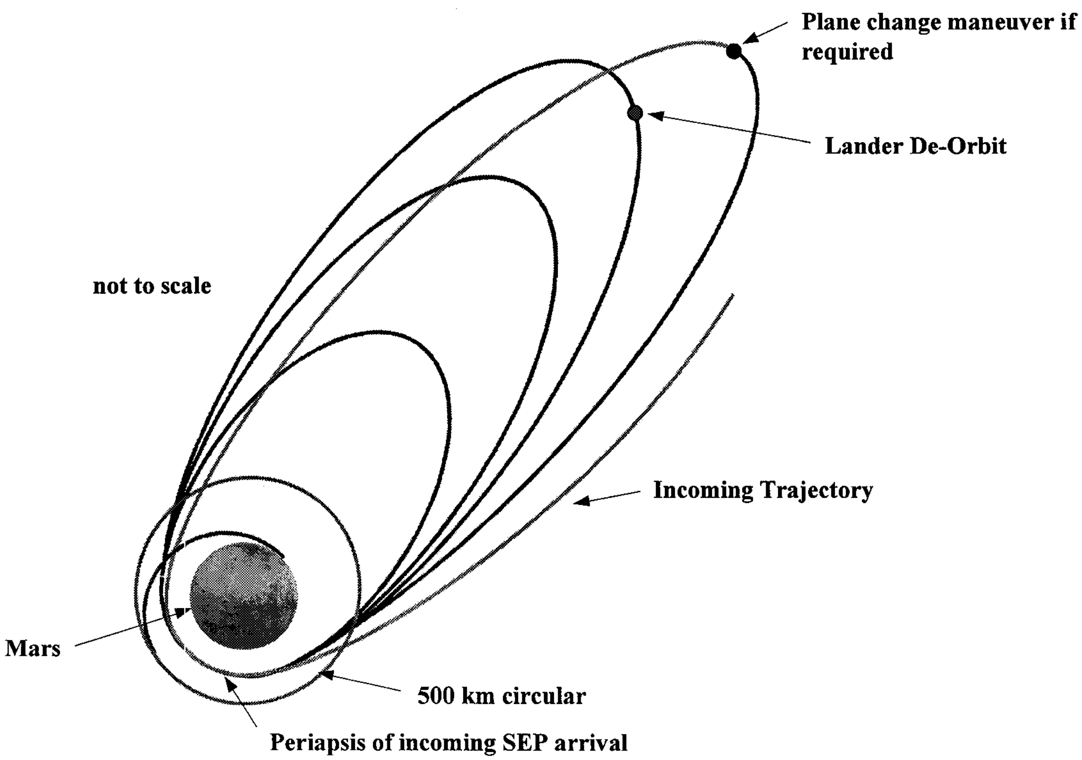


Fig. 6 SEP capture at Mars into a 5-day elliptical orbit and subsequent spiral down to a 500-km circular orbit altitude.

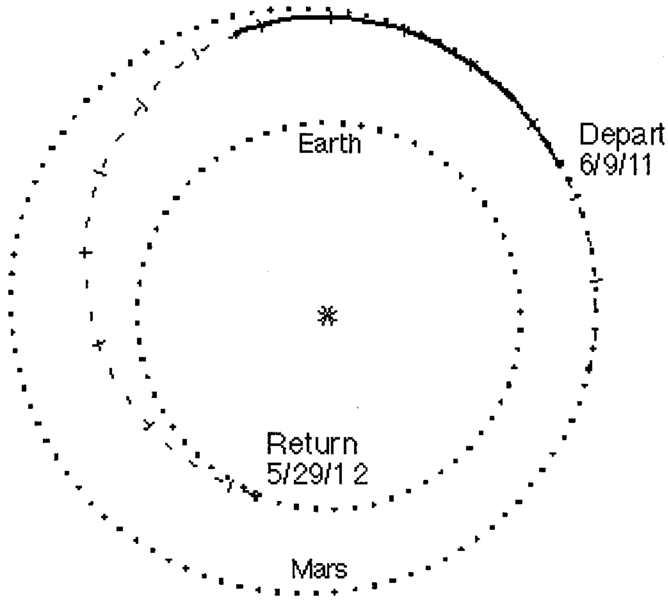


Fig. 7 Return trajectory from Mars to Earth following the SEP spiral to Mars escape.